STEREO Mission Design

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The Goddard Space Flight Center Integrated Mission Design Center (IMDC)

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1 OVERVIEW

This document summarizes the work of the GSFC STEREO Mission Definition Team to provide a detailed design definition for the STEREO Mission. In addition to the team efforts, the IMDC process further refined this design. The starting point for the team's efforts is the "SOLAR TERRESTRIAL RELATIONS OBSERVATORY (STEREO) MISSION CONCEPT, a case study to demonstrate feasibility" document, written by Jim Watzin/GSFC and Joe Davila/GSFC. Significant portions of this document are included as an appendix in "THE SUN AND HELIOSPHERE IN THREE DIMENSIONS, Report of the NASA Science Definition Team for the STEREO Mission", which can be found at http://sd-www.jhuapl.edu/STEREO/. The Science Definition Team Report provides details on the STEREO science objectives.

The STEREO spacecraft is configured using many hardware and software elements with demonstrated protoflight and flight proven performance. The Small Explorer (SMEX) program SMEX and SMEX-Lite architecture are cited in the previous study as a proof of feasibility. However, this study makes no presumption as to how or where the spacecraft is produced, but only presumes ready availability of this class of technology and the acceptance of aggressive project management and systems engineering techniques.

A straightforward instrument to spacecraft interface simplifies the payload integration and observatory-level test sequence. The integration and test of the instruments and spacecraft will require no new technology in order to produce the flight ready article. Due to the constrained cost for the implementation phase ($120M, both spacecraft), a single string/limited redundancy spacecraft bus approach is sufficient to meet the mission objectives while minimizing the overall program cost.

2 MISSION ORBIT

The STEREO mission requires observations from two spacecraft with varying degrees of angular separation within the ecliptic plane. Although there is an optimum viewing geometry for the proposed science instrumentation package, good science can be obtained over a wide range of spacecraft angular separation. With this in mind, a slowly drifting heliocentric orbit, which can be reached by direct launch vehicle insertion, is the optimal low cost solution. This option allows us to avoid the mass increases and spacecraft complexity associated with maintaining the spacecraft in a fixed position in space. This approach has the added benefit of only requiring two spacecraft configurations (or operating conditions) during the mission - the launch configuration and the deployed configuration. With no intermediate orbit transfer or parking configuration and no station keeping requirements to be met, the spacecraft design becomes very straightforward.

The heliocentric, "drift-away" orbit which was chosen has an energy requirement of approximately 0.78 km^2/s^2. A C=1.0 km^2/s^2 was used for launch vehicle performance analysis. One spacecraft would be placed in a leading trajectory ahead of the Earth in its orbit and the other would be placed in a lagging trajectory following the Earth in its orbit. The first spacecraft will drift away at 15°/year and the second spacecraft at 30°/year – this will minimize the drift of the first spacecraft in the event of any launch slips for the second spacecraft. See Figure 1, STEREO Orbit and Figure 2, STEREO Range History.

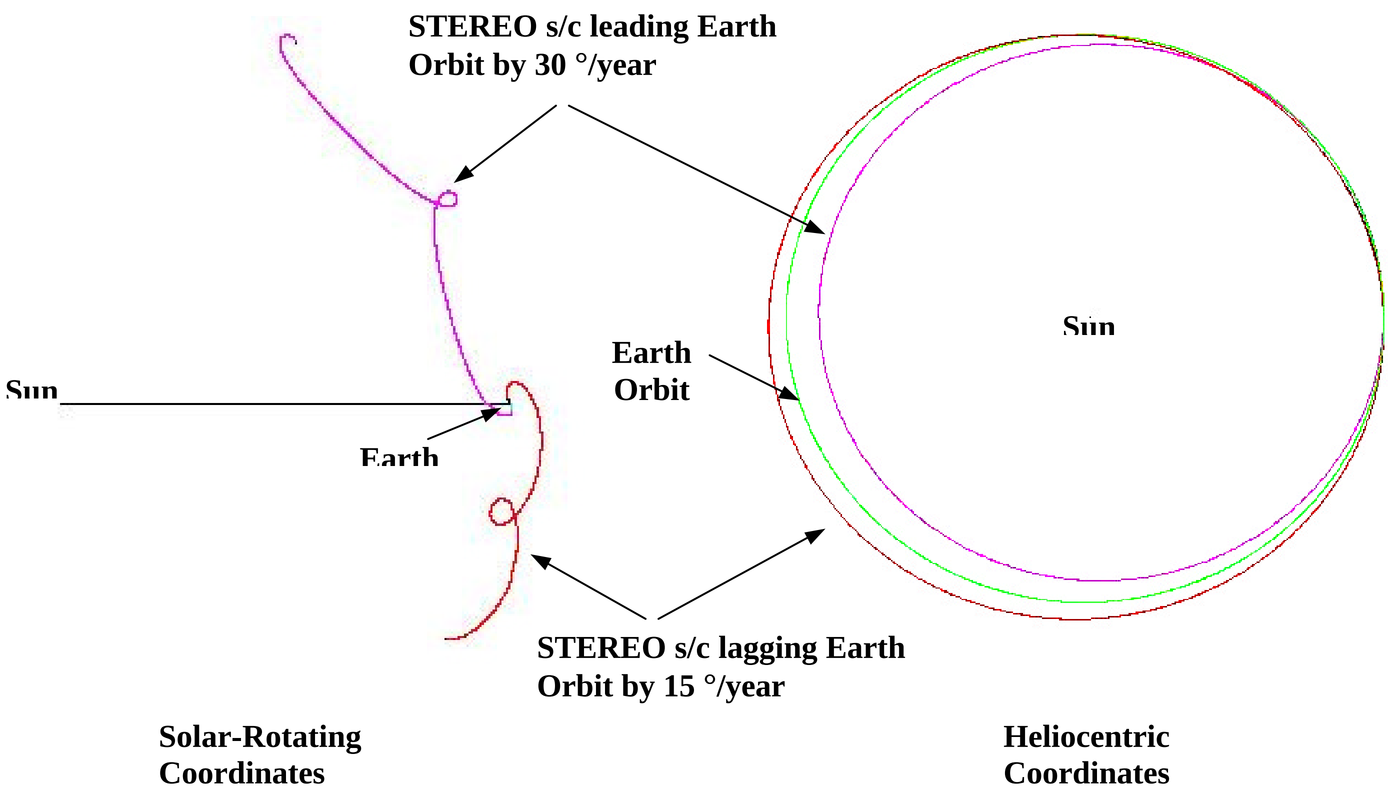


Figure 1: STEREO Orbit

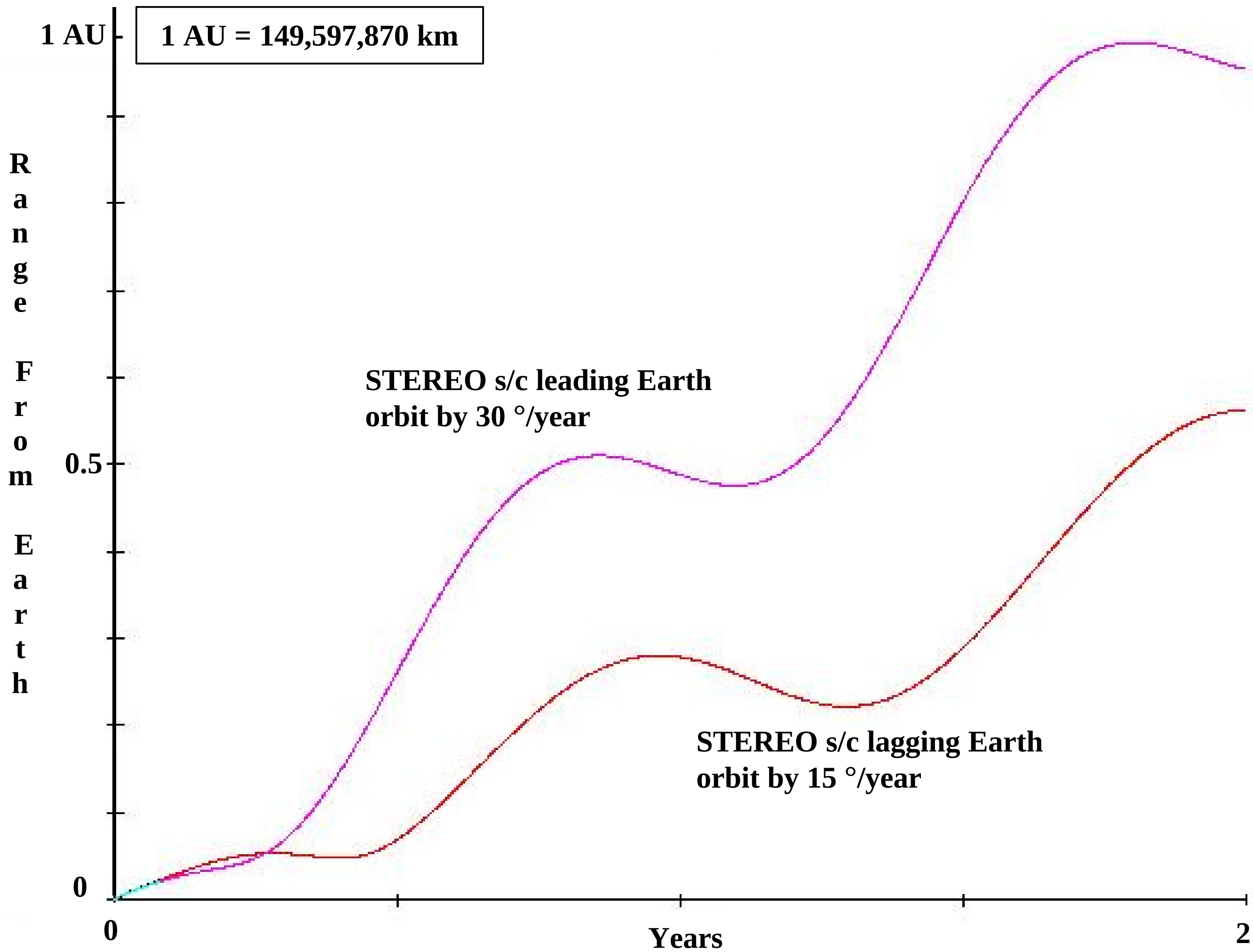


Figure 2: STEREO Range History

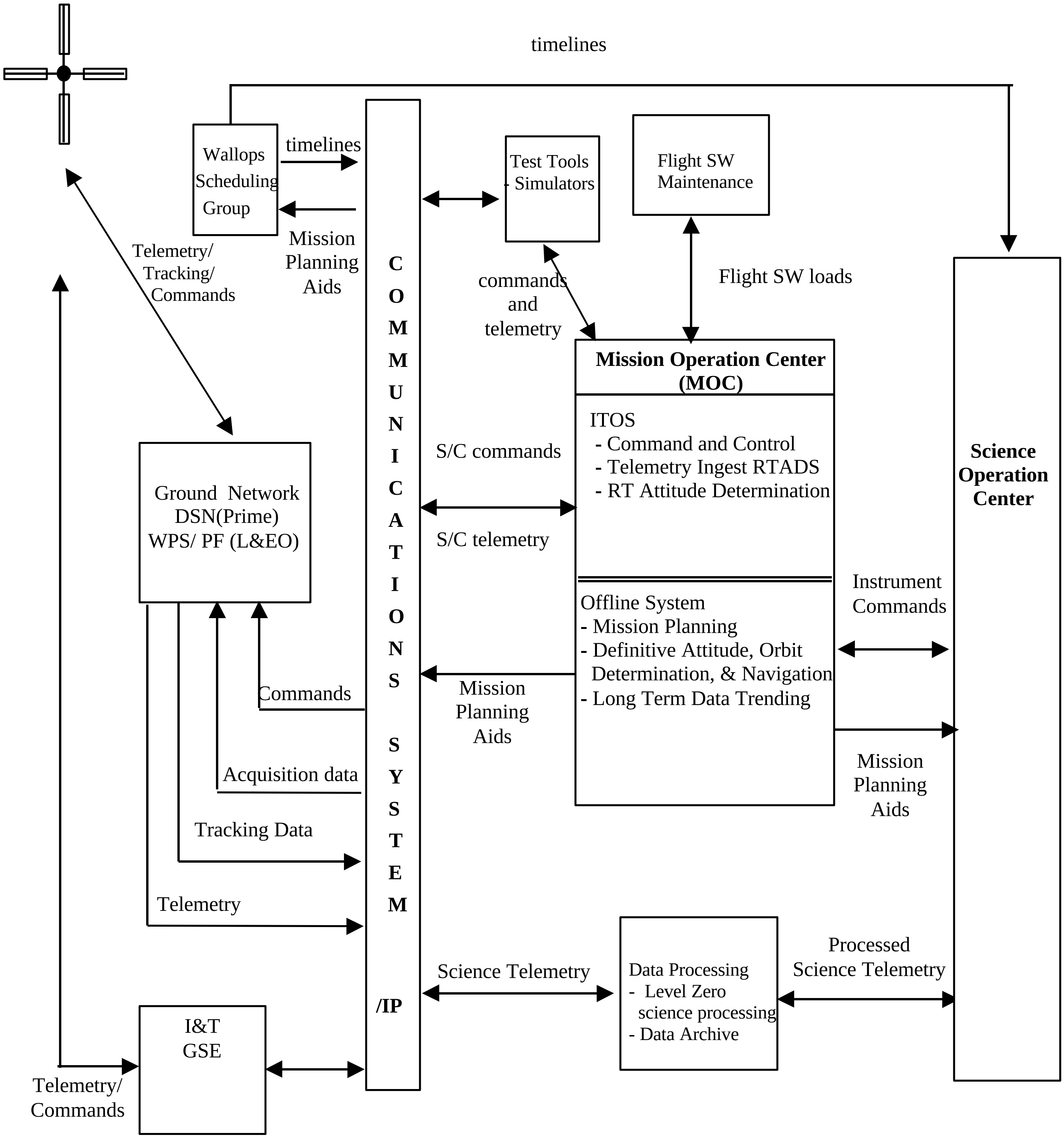


Figure 3: STEREO Proposed GDS Architecture

3 Launch Vehicle

A number of launch vehicle options have been evaluated in varying degrees of detail for the STEREO mission. The launch vehicles evaluated for this mission are briefly discussed below.

3.1 NASA Shuttle

Launch of both STEREO spacecraft as a combined payload on the Shuttle appears to be very feasible. In order to fabricate and qualify a flight carrier and to meet shuttle safety requirements, a very rough estimate of cost to the STEREO project is on the order of $25M. Because of the perceived difficulty of being manifested on the Shuttle and lack of time to pursue this option in detail, the shuttle was not chosen by this study as a launch option.

3.2 DELTA

The DELTA vehicle does not have the capability to provide separate 3rd stages to multiple payloads; consequently, one or both of the spacecraft would require additional propulsive capability to achieve the desired C_3. The DELTA 7326 could lift a 600 kg combined payload mass into a transfer orbit, dropping spacecraft #1 there, then boosting spacecraft #2 into its heliocentric orbit ($C_3 = 1.0$) using a STAR 37 upper stage. After about half an orbit, Spacecraft #1 would reach the proper injection point at which time it would use its on-board propulsion system to boost to the heliocentric orbit. In this scenario, one spacecraft would require a much larger on-board propulsion system (larger tanks, larger thrusters or both), compromising the desire for identical spacecraft weight and jeopardizing mass margin.

In an alternative scenario, a 2-stage DELTA could be used and each STEREO spacecraft would have its own Star 30C stage. This approach would require development of new attach hardware for the launch vehicle and provision for the small solid motor on each STEREO spacecraft. The sequence of events for injection to the heliocentric orbits would be similar to what is described above. A good estimate of the mass of the new hardware to be developed would be needed to ensure adequate mass margin.

Because of the requirement for additional design effort, potential for additional cost (over the launch vehicle cost) and the risk to the mission if the DELTA were to malfunction, the option to use the DELTA launch vehicle was discarded. It should be noted that the cost of one DELTA vehicle is cheaper than two SELVS II – type vehicles; however, the non-recurring costs for new attach fitting hardware and cost for additional propulsive capability for one or both spacecraft could result in little or no cost saving.

3.3 ARIANNE 5

The Arianne 5 offers the advantage of significant cost savings over any other ELV option, with a launch cost on the order of $1-2M. Because STEREO would be a secondary payload, it is not known if one or both spacecraft would be launched at the same time. The information provided for accommodating auxiliary payloads on the Arianne indicates that STEREO could not be accommodated – the payload attach structure provides for up to eight, 80-100 kg payloads. The total mass for two STEREO spacecraft could be accommodated, but each spacecraft exceeds the individual payload specification for the attach structure.

If a STEREO-unique attach structure could be provided to allow the use of the Arianne, the STEREO spacecraft could be injected into the proper orbits. This option would be very similar to the launch scenario and requirements for the DELTA. However, because STEREO would be a secondary payload, there is no guarantee that each spacecraft would be dropped off where it could be directly injected into the proper heliocentric orbit. Depending on the drop-off point, one or both of the STEREO spacecraft could be required to orbit for anywhere from a half an orbit (no delay) to 1.5 years (worst case delay) before reaching the proper injection point. If any significant time is required in orbit for either spacecraft, this would impose an additional burden to survive eclipse periods. In addition, there is the increased cost and complexity of launching on a foreign provided launch vehicle. Because of these factors and the lack of time under this study to explore the options in detail, the Arianne 5 launch vehicle was not chosen for the STEREO mission.

3.4 TAURUS

The four-stage TAURUS with standard motors and a 63-inch fairing has a capability to C_3 = 1.0 km^2/sec^2 of 289 kg. During the course of the study, however, it was determined that the 63-inch fairing was not large enough to accommodate the spacecraft. A 92-inch fairing is available for use with the TAURUS, but this reduces payload capability by 13 kg, which results in <u>negative</u> weight margin. Additional performance can be attained by using the XL motors (343 kg for C_3 = 1.0, 92-inch fairing), but at the current time, the TAURUS XL configuration is not approved for use by NASA missions.

3.5 ATHENA II

The Athena II with a 92-inch fairing has a capability of 295 kg for C_3 = 1.0, resulting in positive weight margin for the mission design. However, there are volume constraints that makes this option unfeasible for the spacecraft design. Both the spacecraft and STAR 37 motor do not fit within the available fairing volume. Therefore the Athena II is not a viable option for STEREO.

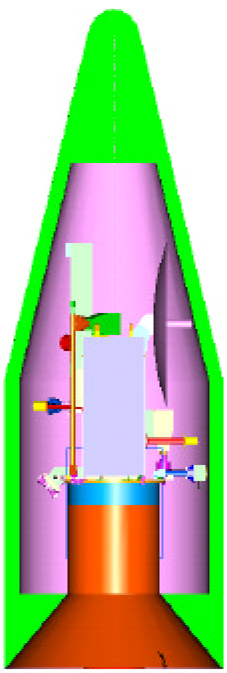
3.6 Conclusion

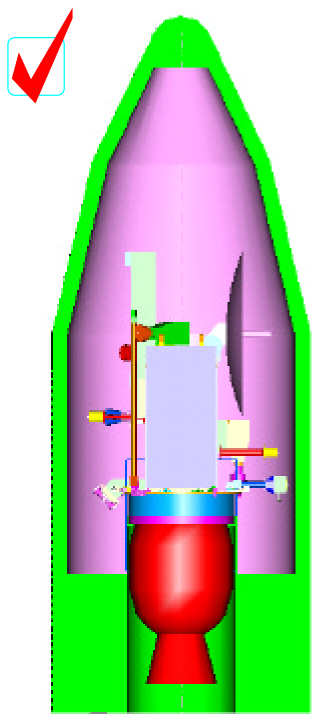
In the launch configuration, this mission clearly will not fit in the Athena II/ Star 37. The Taurus 92/Star 37/ Orion 38 can accommodate this mission from a volume stand-point, however, as shown on Fig. 4, the lift-off mass margin is a negative 13 kg. A possible solution would be to look for weigh savings in the structure, subsystems or payload area's. There is always the option of using the DELTA-Medium-Lite (3m Diameter, 3 Stage /Star 37 FM), see Fig.4. This vehicle will easily accommodate both mass and volume requirements, however it is a costlier option.

ATHENA II- 38 Model
l not meet mass and volume
requirements

Taurus 92/ Star 38/ Orion 38
Meet’s volume requirements,
has a -negative mass margin*

Delta II Medium Lite, 7326
Mass and volume ‘s are meet,
but costlier launch vehicle





*Note- With some re-design these
mass margin’s may be meet

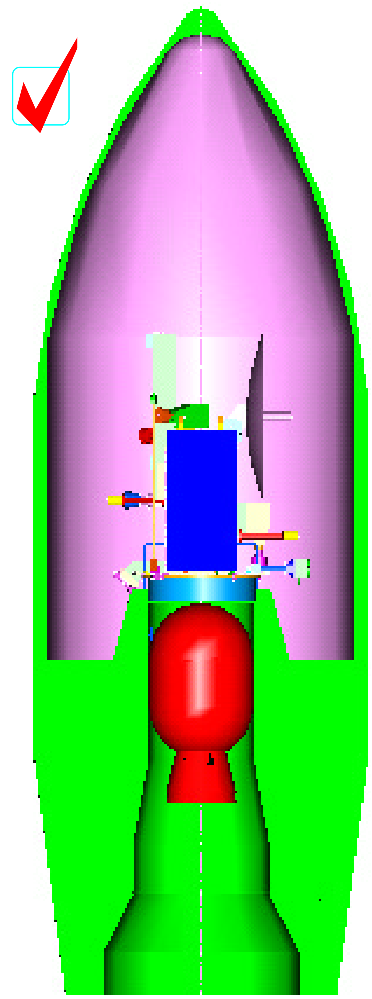


Figure 4: STEREO Launch Vehicle Trade-off

4 MISSION OPERATIONS

This study assumes that the STEREO mission will be based on ground and data systems similar to those used for TRACE and WIRE. The spacecraft and instruments will function autonomously for the majority of the mission, with mission operations highly streamlined after initial checkout and calibration. DSN contact time can vary over the life of the mission and will generally be shorter early in the mission, up to a maximum of 2 hours/spacecraft (34m) at end of life (1AU). The location of the control center is somewhat arbitrary when considering the advancements in telecommunications. Communication services between the Jet Propulsion Laboratory (JPL/ DSN), the primary source of data, and GSFC are already in place and would simply require bandwidth allocation. If the control center were located outside of GSFC, a connection through GSFC to JPL would be required. GSFC Flight Dynamics Facility (FDF) would provide tracking and navigation services which would allow STEREO to take advantage of the PC based systems being developed for TRACE and WIRE.

The primary ground control requirements can be met by using the Integrated Test and Operations System (ITOS). See Appendix A. ITOS has 7 years of SMEX heritage and takes advantage of cost effective automation features. The automation features allow for unattended health and safety monitoring, data processing, and anomaly notification. Operations will be reduced primarily to mission planning and post-pass data evaluation, which will enable operations support to be constrained to a standard 40-hour week. Operations of both STEREO satellites must be done under the same observing program to maintain the science requirement (Images from the two spacecraft must be taken within 1 second apart).

The spacecraft will have selectable telemetry rates to accommodate link margin restrictions, which will give the operations personnel considerable flexibility in mission planning. The DSN 34 meter system is the baseline for the STEREO communication system; however, in times of conflict the 70 meter system can be used to dump science data faster and to conserve spacecraft power. Since both satellites will not be in view from the same ground station at the same time, multiple DSN station contacts will be needed to support command uploads to the spacecrafts. This is particularly critical when responding to an activity alert from "Beacon Mode. Command uploads will consist primarily of station contacts and science data dump commands. Event driven commands will be available for synchronized observing modes. Optical ephemeris updates might require to be loaded more often than weekly to maintain the spacecraft antenna pointed to the ground station within adequate coverage. Ephemeris updates shall be provided by the Flight Dynamics Support System located at GSFC. Daily onboard clock adjustments might be needed depending on the spacecraft time distribution design implemented (oscillator vs. standard frequency vs. ultrastable oscillator). A Command Management System (CMS), similar to the TRACE system, will take ground station contact schedules and science command inputs to produce spacecraft command loads. Because of the increasing distance between the STEREO spacecraft and earth as mission time progresses, real-time commanding will become increasingly difficult and should be constrained to early orbit checkout activities and to other times when absolutely necessary – anomaly situations or events of extraordinary science interest.

4.1 BEACON MODE "Space Weather Forecasting"

A goal of the STEREO mission is to implement a "Beacon Mode", which would provide near real-time notification of significant solar events with potential impacts to the earth and near-earth environments. On-board instrument event recognition could identify these events as they develop and trigger the "Beacon Mode" that would attempt to call for ground support. This mode would initiate a request for ground support by sending a low rate beacon to scheduled listening stations. Once the signal is detected the station would activate a paging request to the operations personnel. The flight operations team would then schedule a DSN station contact and issue the data collection commands. This concept does have drawbacks in that it would require constant ground monitoring. A study by an outside contractor has been initiated to develop a simple, low cost concept for monitoring and receiving the beacon mode signal.

5 INSTRUMENTS

The mission definition efforts have used the same instrument complement defined in the Watzin/Davila feasibility study, plus one more instrument added by the STEREO Science Definition Team, the Heliospheric Imager, for a total of six instruments. This instrument complement will be identical for each spacecraft. It should be kept in mind that these are strawman instruments, and the actual instrument selection will be via the NASA AO selection process, directed by NASA Headquarters.

In the process of this study, additional information was obtained concerning instrument designs and resources (mass, power, data rate). In the case where an instrument was based on a previously proposed instrument, the proposing investigator was contacted to obtain additional and/or more accurate information than was available from the previous study. In the case of the referenced heritage instruments, the original PI or a member of the PI team was contacted to obtain additional detail on the heritage instrument. See Table 1 for a listing of the STEREO instruments.

Table 1: STEREO Instrument Complement

Instrument	Heritage	Measurement
Solar Coronal Imaging Package (SCIP)	Coronograph is similar to the SPARTAN 201 coronograph with offset occulter Coronal Doppler Imager based on simplified version of TRACE and EIT	White light imager w/CCD detectors EUV Doppler imager
Energetic Particle Detector (EPD)	ACE	Absolute intensity and energy spectra of energetic particles
Radio Burst Tracker (RBT)	Similar to WIND and GALILEO	Tracks solar radio disturbances
Magnetometer	Similar to GIOTTO, CLUSTER, Mars Global Surveyor	Interplanetary magnetic fields
Solar Wind Plasma Analyzer (SWPA)	WIND	Proton and electron density
Heliospheric Imager (HI)	New design	Tracks the solar wind from the sun to the earth

In order to keep the instrument to spacecraft interface as simple as possible, if data compression is required for instrument data, it will be done by the instrument.

Instrument data rates will be presented in the communications subsystem section.

6 SPACECRAFT

The STEREO bus is configured as a zero-momentum, three-axis stabilized spacecraft with two, and bi-fold solar array panels for power generation. Each spacecraft orbits the Sun with the respective SCIP instrument boresight always pointed towards the sun. The stowed spacecraft concept fits within the Taurus XL 92-inch launch vehicle fairing with adequate margin as shown in Figure 5. The fully deployed, on-orbit configuration is shown in Figure 6. The spacecraft provides a power interface for each instrument and uses a MIL-STD-1553 bus architecture for communications between the spacecraft computer, the instrument, and specified bus subsystems as shown in the block diagram presented in Figure 7.

The spacecrafts shall meet a total mission life radiation dosage of 30 K rads. They also shall be able to withstand damage caused by Single Event Effect. Techniques such as latch up detection circuitry, error correction software and adequate part selection will be used to safeguard the spacecrafts from the radiation environment.

6.1 Mechanical/Thermal

The spacecraft bus is a square structure, 0.6 x 0.6 x 1.2 meters. The spacecraft uses a 38” separation system adapted to the selected launch vehicle. The core structure is made up of composite and aluminum honeycomb panels, truss, ring fittings, brackets diagonals and stringers. The bus is considered a closed architecture configuration, with subsystems mounted internally on structural members. Instruments are all mounted on the exterior of the bus.

6.1.1 Mechanisms/ Deployables

The solar array consists of two sets of bi-fold array panels, one on each side of the spacecraft (+Y, -Y). The arrays are stowed against the sides of the bus and are held using a non-explosive actuator to minimize deployment shock. Because the normal orientation of the spacecraft is towards the sun, the arrays have a fixed orientation and no array drive is required.

The High Gain Antenna (HGA) is gimbal mounted and can pivot back up to 45 degrees.

The Magnetometer is mounted on a 6 meter, multi-fold boom; deployment is initiated using a non-explosive actuator to minimize deployment shock.

The Radio Burst Tracker instrument utilizes 3 wire antennae (one in each axis) as detectors. Each wire (10m) is contained within and deployed by an OSC model 625 Hingelock TEE deployer. This is a deployment only mechanism and has no retraction capability.

6.1.2 Thermal

A reasonably simple thermal design is achievable due to the stable thermal environment (deep space, sun pointing) and the temperature limit requirements of the bus components.

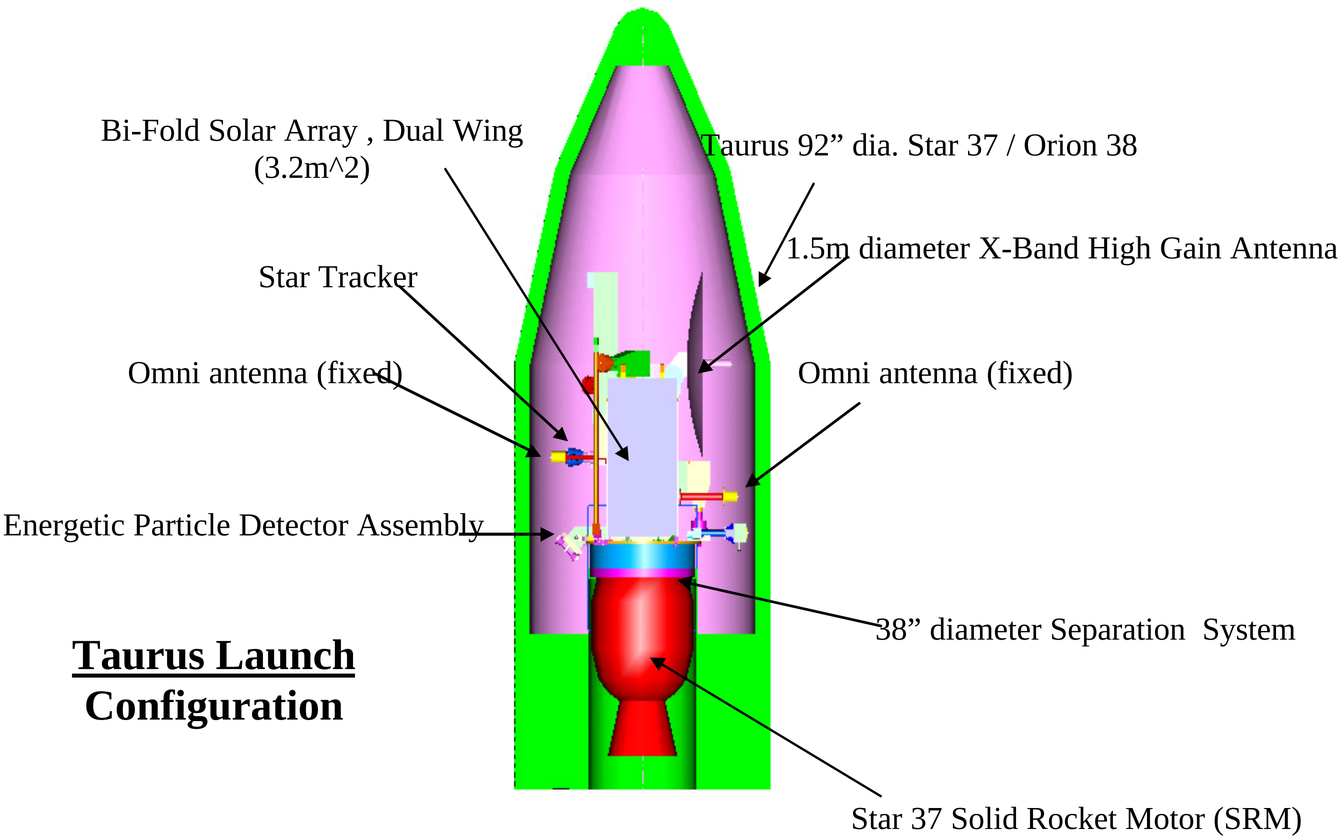


Figure 5: STEREO Launch Configuration

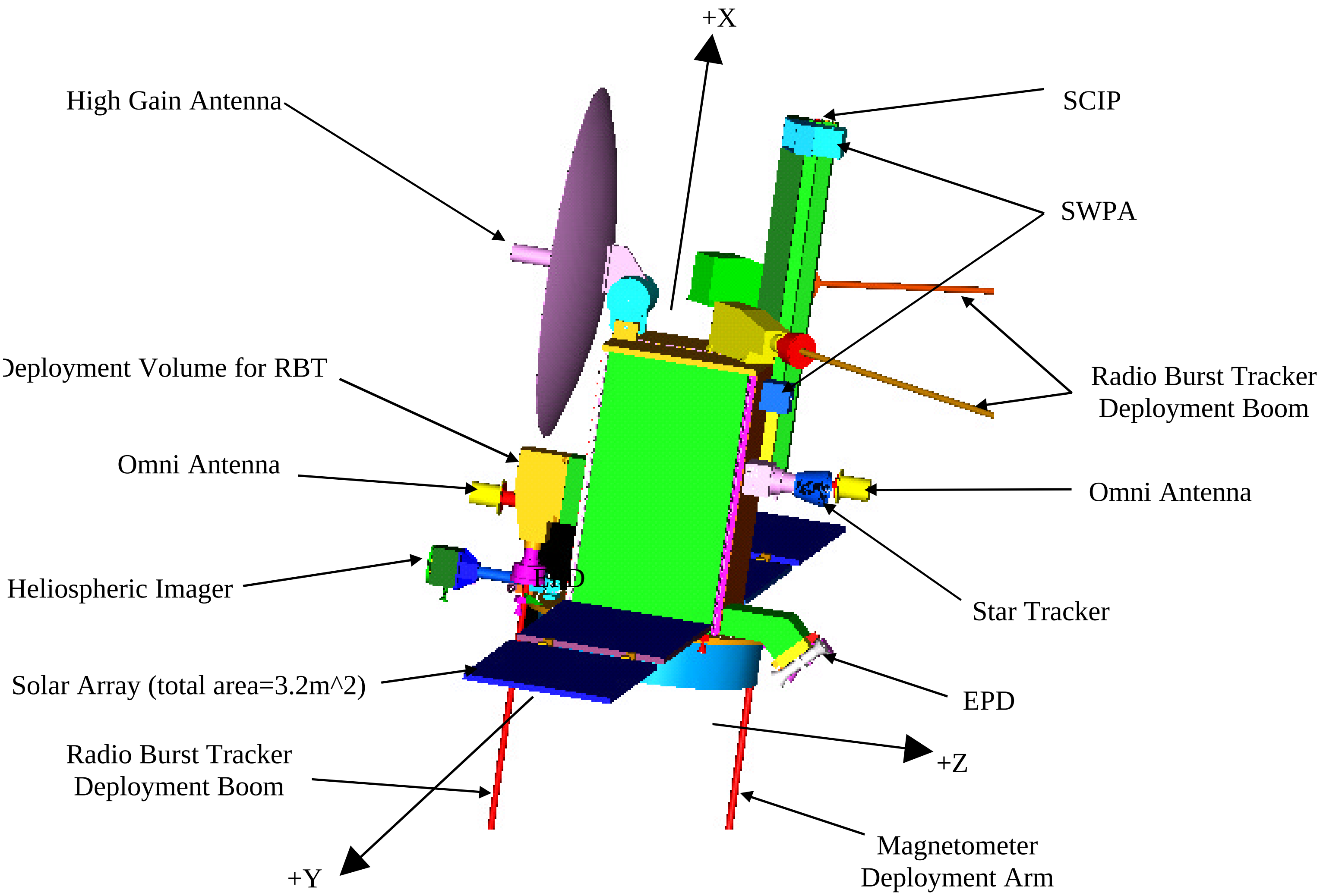


Figure 6: STEREO On-orbit Configuration

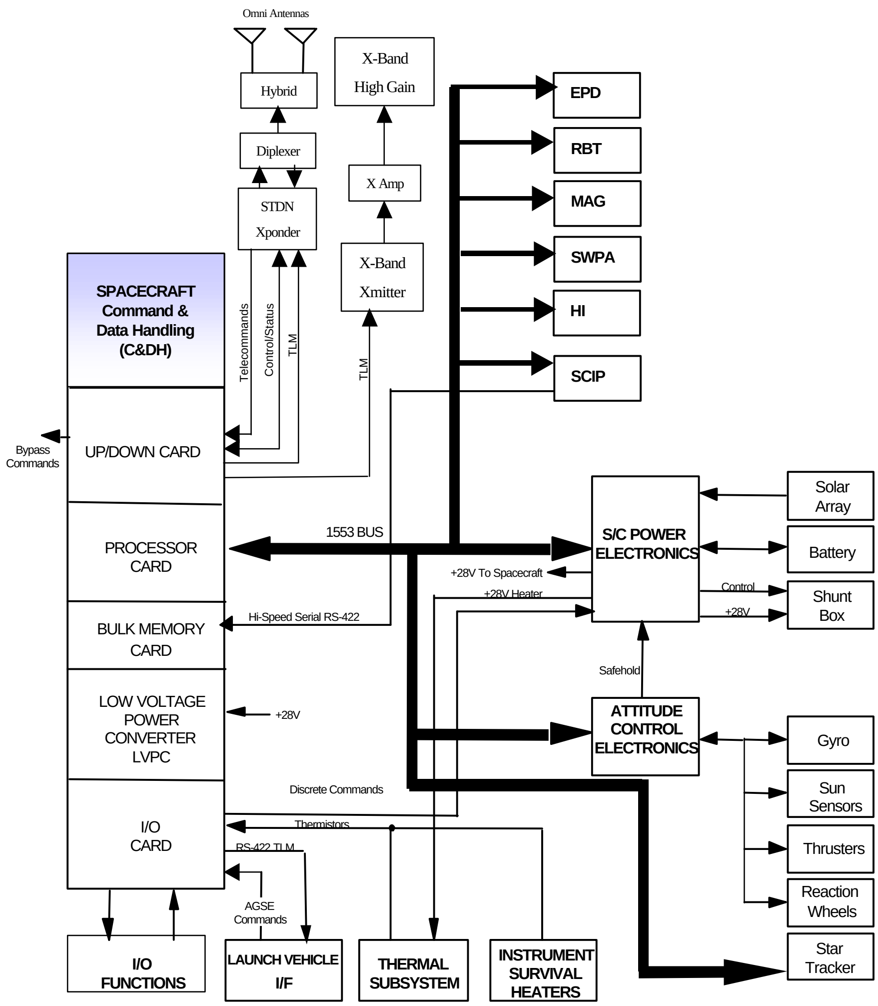


Figure 7: Data System Functional Block Diagram

The COMM subsystem equipment accommodation is the major thermal system driver for the bus. The COMM equipment (S-band transponder, X-band Amplifier (TWTA), and X-band Transmitter) is mounted on a heat pipe panel on the “startracker” side of spacecraft, with thermal louvers covering the radiator panel (aluminum honeycomb with imbedded heatpipes). Fixed conductance heat pipes are required to spread heat effectively over the entire radiator – the TWTA dissipates 90 watts thermal (heat) energy in a footprint of ~ 6 cm x 1.2 cm. Because the COMM system only operates in its high power mode (~ 180 watts) 2 hours per day (max) the louvers covering the radiator will effectively “modulate” the radiator area to allow long term operation of the COMM equipment without the need for heater power when the equipment is in standby/off (5 watts). The louvers are “vane” type, used on many previous missions (ATS-F, MMS, ERBS, XTE, TRMM, L7) over the last 25 years, with a “passive” design: bimetallic actuators set blade angle in direct response to base/equipment temperatures; the multiple blade design is somewhat self-redundant.

Except for the COMM equipment, the thermal design utilizes fixed radiators with active heater control where necessary. The battery is thermally isolated from the main spacecraft with a passive radiator and active heater control to maintain its narrower temperature range. No operational heaters are required for other spacecraft bus equipment.

The Magnetometer, SCIP, HI and EPD will be thermally isolated from the bus due to location and/or cooler instrument control temperature ranges. The RBT and SWPA instruments are thermally coupled to the spacecraft.

The SCIP, HI and EPD instruments will utilize cold-biased radiators with active heater control to maintain their narrower temperature range. The SCIP and HI detectors (CCDs) will have dedicated radiators to cool them to the –70°C operating temperature. The Magnetometer will be completely covered with multilayer insulation (MLI) and utilize a small heater for thermal control.

The instruments will require 55.5 watts of heater power for normal operations and the battery requires 6 watts for a total of 61.5 watts of normal mode heater power. During the initial checkout period, the instruments will require 143 watts of heater power for bake-out.

+X surfaces and all surfaces not used as radiators are covered with MLI; +X (sun face) MLI has an outer layer with low solar absorptance coating.

6.2 C&DH Subsystem

The C&DH Subsystem consist of three cards, an Uplink /Downlink card, a processor/memory card and a housekeeping card (I/O CARD on Fig.7). The C&DH subsystem will utilize a 1553 Bus as the interface. The prime communication path between the instruments and the Spacecraft processors is the MIL-STD-1553B Data Bus. All onboard data is transmitted via this data bus, except for the SCIP image data, which will be sent via the high speed serial RS-422 interface. The SCIP image data interface to the system will be via an RS422 interface on the C&DH system, as the higher data rate of the SCIP exceeds the capability of the 1553 bus. The 1553 Bus will be the interface between the processor card (acting as the BC) and the Uplink/Downlink card, the Solar Coronal Imaging Package (SCIP) (Engineering Data only), the Energetic Particle Detector (EPD), the Heliospheric Imager (HI), the Magnetometer, the Radio Burst Tracker (RBT), and the Solar Wind Plasma Analyzer (SWPA). There is an interface consisting of Clock and Data interfaces from/to the Uplink/Downlink card to the receiver/transmitter of the Communications Subsystem. Also, there are various thermistor and current/voltage monitoring from the housekeeping card.

The CD&H system will manage the spacecraft and instrument data collection, command contribution, bulk telemetry data storage with EDAC.

The Mongoose V spacecraft processor card is the central processor for the C&DH/ACS of the STEREO spacecraft. The Mongoose V processor card is based around the Mongoose V 32-bit Rad-hard RISC Processor, which is operating at 12 MIPS. The Mongoose V card includes the following:

- 4Mbytes of jumper-configurable EEPROM that can be configured (in 64Kbytes steps) as non-writeable bootstrap EEPROM, write-protected EEPROM, and flight-writeable EEPROM;
- 8 Gbits of DRAM used for processor local memory. This memory will mostly be used for data storage with a relatively small amount of memory being used for running the processor;
- 1553B Bus controller interfaces
- Spacecraft Time Keeper, which maintains Mission Elapsed Time in seconds and sub-seconds;
- Watchdog timer circuit, which provides a timed reset capability;
- External Timer, which provides a timed interrupt capability;
- Medium Speed Serial Port to transfer serial data;
- External Waitstate Generators;
- RS-422 interfaces.

All the processing functions of the C&DH flight software and ACS flight software are performed on the processor card.

The Uplink/Downlink Card will interface with the receiver and transmitters. The card will perform command synchronization and decoding, and will provide spacecraft clock. The downlink card will generate CCSDS transfer frames and encodes the science and housekeeping telemetry. It also will provide a synchronous medium speed serial input port to acquire the science telemetry.

The Housekeeping card (IO CARD) will perform the Health and Safety monitoring of the spacecraft. This includes the monitoring of passive telemetry points such as temperature sensors, and actives such as currents, voltages, solar array load cells among others.

The system shall allow for a high-speed serial port for the SCIP telemetry data.

6.2.1 Timing

The images and data from both spacecraft shall be taken within 1 second of each other in order to allow for data triangulation processing.

The C&DH system will be responsible to provide a clock signal. A simple oscillator, rather than a high precision frequency standard will meet this requirement.

6.3 Power Subsystem

Power is generated using two solar array strings with a total combined area of 3.38 square meters (2.88 sq. meters active area of triple junction GaAs cells). Use of standard, single junction GaAs cells would require a total of 4.69 square meters of area, increasing the complexity of the mechanical design and adding 5.6 kg of mass. This array will yield a higher power density than with either standard GaAs cells or Si cells, in addition to exhibiting lower radiation degradation and temperature dependence than a traditional silicon array. The array will supply about 700 W of conditioned power at end-of-life (EOL). Conditioning is achieved with power supply electronics (PSE) based on the MAP design and sized up to a maximum 800 watt load. Energy storage is achieved with a single 26.5 Ah Lithium ion battery. The solar arrays are sized to provide 100% of the power required over the nominal mission life; the battery is sized to provide power during the 120 minute launch and initial acquisition period where power is required for communications and to null launch vehicle tip-off rates and properly orient the spacecraft. A single bus (28 ±7 V) supplies power to both the bus subsystems and the instruments. STEREO component mass and power are shown in Table 2. Figure 8 shows the power load profile for early orbit and normal mode operations.

Table 2: STEREO Mass and Power

Component		Qty	Mass Total (kgs)	Size (l,w,h) inch	Power Total (watts)
SPACECRAFT					
ACS	Star Tracker	1	4.99	6 x 6 x 8	13.00
	IRU	1	5.53	12.9 x 8 x 3.75	39.00
	CSS	4	0.82	1 x 1 x 1	0.00
	Reaction Wheels	4	13.79	4.1 dia x 2	36.00
RCS	GN2 tank	1	11.34	18 dia x 26 lg	0.00
	HP transducer	1	0.45	3 dia x 6 lg	0.50
	LP transducer	1	0.45	3 dia x 6 lg	0.50
	fill & vent valve	1	0.23	1.5 dia x 3 lg	0.00
	filter	1	0.11	1 dia x 1 lg	0.00
	regulator	1	1.36	6 x 6 x 12	0.00
	thruster	6	0.27	1.5 dia x 3 lg	0.00
	Brackets, plumbing and misc.	1	4.54	a/r	0.00
	propellant		18.00		0.00
CDH	Computer	1	8.00	8.7 x 9.2 x 12.2	30.00
POWER	Battery		9.90		0.00
	Solar Array		17.50		0.00
	Power Supply Electronics (PSE)		14.00		40.00
COMM	1.5m HGA	1	6.53	58.5 dia x	0.00
	HGA gimbal	1	8.62	7.4 dia x 9.4	0.00
	Omni antenna	2	0.91	3.8 dia x 5.2	0.00
	X-band TWTA	1	0.86	13.25 x 2.46 x 2.34	13.30
	S-band Transponder	1	3.63	7.8 x 4.8 x 5.8	5.00
	X Band Transmitter	1	3.63	7.9 x 6.5 x 2.8	4.20
	Coax	1	0.50	A/R	0.00
	Turnstile	1	2.31	4.0 dia x 6.0	0.00
THERMAL	Blankets, heaters, etc	A/R	3.00		61.50
STRUCTURE	Bus structure		36.30		0.00
	Instrument support structure		6.00		0.00
	Misc. hardware		5.40		0.00
	Harness		4.10		0.00
	Magnetometer Boom	1	3.50	1.25 dia x 118 lg	0.00
	Spacecraft Total Weight		**196.57**	**S/C Total Power**	**243.00**
INSTRUMENTS					
Energetic Particle Detector	EPD unit	1	1.40	10 x 7.1 x 4.75	2.00
Heliospheric Imager	HI unit	1	6.80	8 x 8 x 4.8	20.00
Magnetometer	Sensor	1	0.30	2 x 2 x 4	0.50
	Electronic Box	1	1.70	7.9 x 6 x 3.2	1.50
Radio Burst Tracker (RBT)	Antenna	3	1	1.12 dia x 400 lg	
	Deployment Mechanism	3	3.30	20.1 x 7.1 x 8.23	
	Electronics	1	4.00	7.9 x 4 x 1.2	4.00
Solar Coronal Imaging Package					
	SCIP Unit	1	30.00		20.00
Solar Wind Plasma Analyzer (SWPA)					
	Faraday Cups	2	4.00		1.00
	Electronics	1	1.00		2.00
	Instrument Total Weight		**53.50**	**Instr. Total Power**	**51**

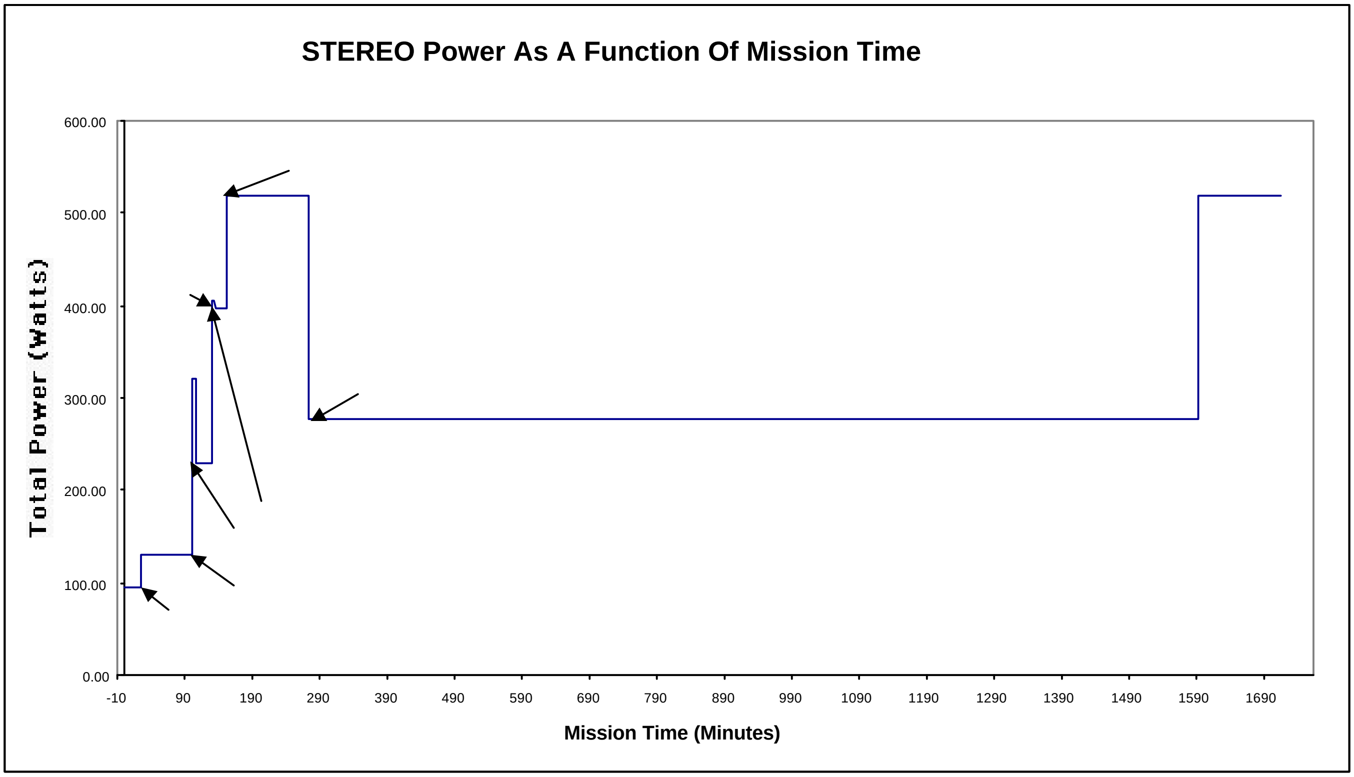


Figure 8: STEREO Power Profile

6.4 Attitude Control & Determination Subsystem

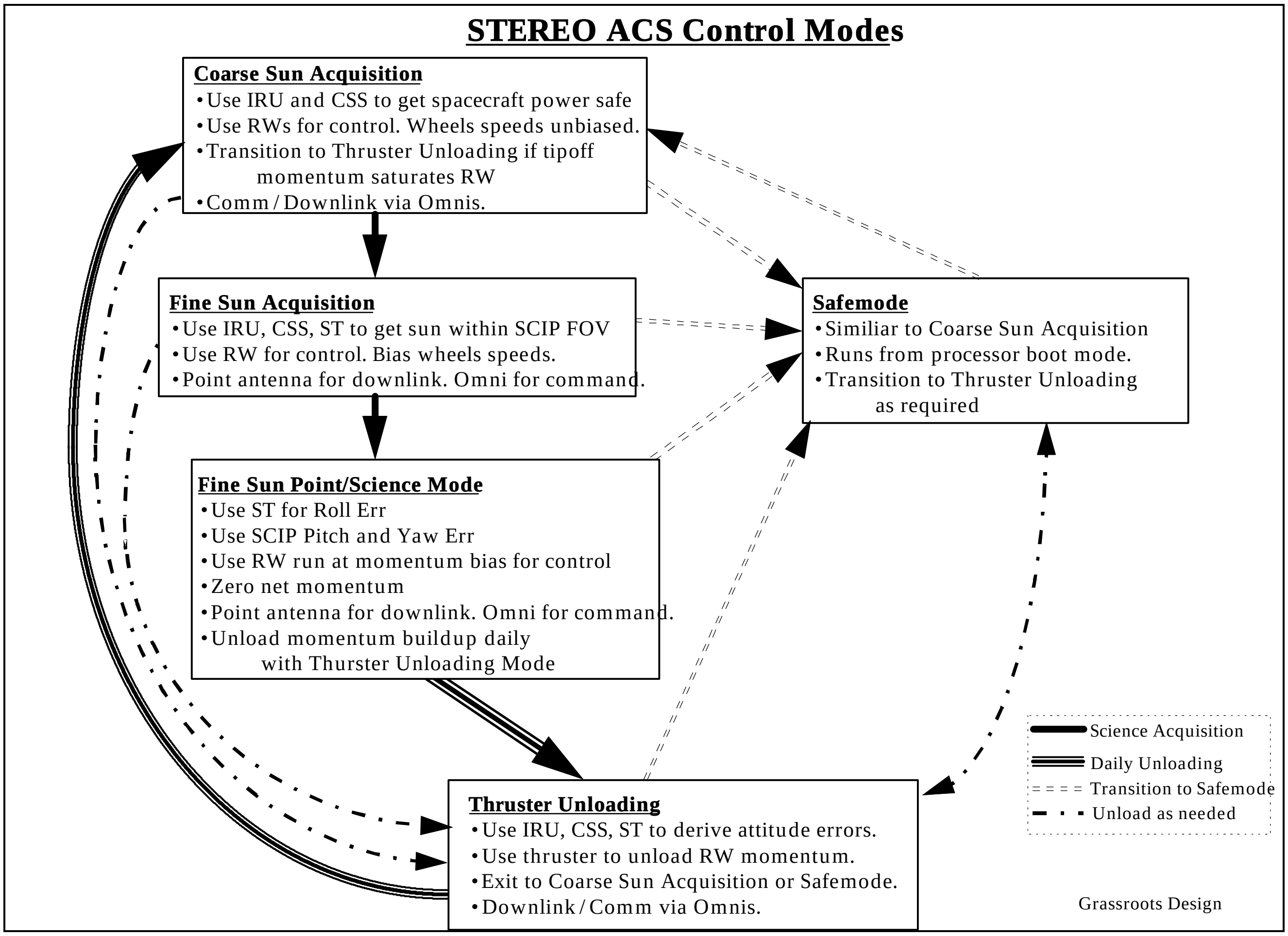
The Attitude Control & Determination System consists of sensors and actuators, controlled by the ACS processor. Four reaction wheels and a set of thrusters comprise the actuators. The sensors consist of an IRU (Inertial Reference Unit- gyro pack), a star tracker, and six coarse sun sensors. An Attitude Control Electronics (ACE) box will provide the interface for all the actuators and sensors, as well as the Safehold function. The ACE box will interface with the spacecraft computer over a MIL-STD-1553 Data bus.

The main driver for the STEREO ACS requirements is the Solar Coronal Imaging Package (SCIP) instrument. The SCIP will provide pitch and yaw position errors accurate to within 0.1 arcsec for fine sun pointing. The ACS must point the SCIP boresight within ±5 arcmin of the sunline before these fine pitch and yaw errors become available. The axis along the SCIP body is referred to as the roll, or X, axis. Pitch (Y) and Yaw (Z) are perpendicular to the SCIP boresight. A star tracker (ST) will provide the roll attitude. With STEREO in a heliocentric orbit, there is no other sensor available to provide the roll attitude. STEREO will also carry a 3-axis IRU to provide rate information and four CSS to provide full sky coverage and position information for initial acquisition. Control is provided by four reaction wheels, which will be run at bias speeds to avoid zero crossings (this will help reduce the jitter generated by the RW), and a cold gas thruster system is used for momentum unloading. The knowledge, pointing, and jitter values for the grassroots baseline system are shown in Table 3:

Table 3: STEREO Pointing Requirements

Knowledge: (3 sigma)	**X** **Y** **Z**	20 arcsec 0.1 arcsec 0.1 arcsec	➔ MAP star tracker accuracy, boresight accuracy ➔ Coronagraph pitch error signal (only once in science mode) ➔ Coronagraph yaw error signal (only once in science mode)
Pointing Performance: (3 sigma)	**X** **Y** **Z**	0.10 degree 20 arcsec 20 arcsec	➔ Required for antenna pointing ➔ Needed to maintain occulter's position if want to consistently study the depth in the corona
Jitter Requirement: (3 sigma)	**X** **Y** **Z**	30 arcsec 1.5 arcsec 1.5 arcsec	➔ Dictated by Doppler Imager: 25% of pixel size = 1σ jitter req ➔ Dictated by Doppler Imager: 25% of pixel size = 1σ jitter req

6.4.1 Control Modes



The ACS control modes are outlined in the diagram above and described in detail in the following section.

Coarse Sun Acquisition

Just after separation from the launch vehicle, STEREO will use the CSS to derive position information and the IRU to detect rates. Based on these errors, the ACS will slew the vehicle to the sun using RW. To maximize the available control authority, the wheels will not be momentum biased during this mode. To minimize wheel size, the RW were sized to store daily momentum buildup, not to handle the maximum separation rates. If the initial tip-off rates are high enough to saturate the wheels, STEREO will use thrusters to unload the momentum, before acquiring with the RW. This will mitigate the risks associated with a thruster acquisition while still minimizing the wheel size. All communications will be handled through the two Omni antennas, since this type of control is too coarse to point the HGA.

Fine Sun Acquisition

Once STEREO has completed Coarse Sun Acquisition, the CSS and ST attitude information will be used to command the RW to place the sun in the SCIP FOV such that the SCIP pitch and yaw error signals are available. This requires positioning the SCIP boresight within ±5 arcmin of the sunline. The IRU will continue to provide rate errors. The HGA will be pointed with 0.1 degree accuracy. Commands are routed via the Omni antennas, but downloads will be through the HGA. At this point, less control authority is required and the jitter must be minimized before entering Science Mode. To accomplish this, the wheels will be commanded to a momentum bias speed. With the sun in the SCIP FOV, the high accuracy pitch and yaw error signals will be available to the ACS and STEREO can proceed to Science Mode.

Fine Sun Point/Science Mode

STEREO will use the SCIP for Pitch and Yaw position errors. The ST will provide the roll error, and the IRU will provide the rate errors. The RW will run at their bias speed to reduce jitter, maintain the required science pointing, and store the daily momentum buildup from the environmental torques. Commands will be sent via the Omni antennas; downloads via the HGA. Once a day, STEREO will transition to the thruster unloading mode to dump the wheel momentum. The daily maneuver will result in about 30 minutes of lost science.

Delta-H/Thruster Unloading

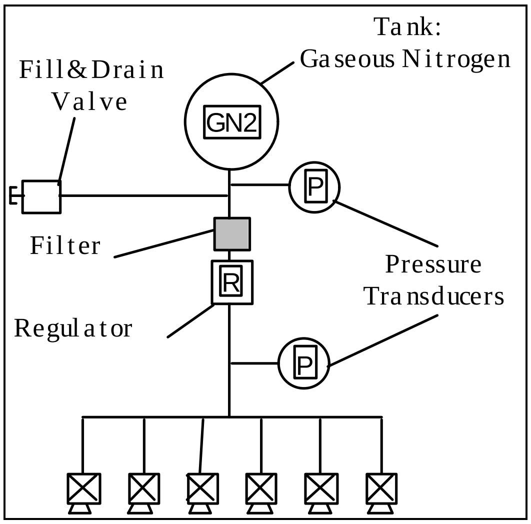
This mode will derive errors from the IRU, CSS, and ST while firing thrusters to unload the spacecraft momentum. This mode will be used for daily momentum unloading, and, in a contingency situation, to unload high tip-off momentum. With thrusters, the ACS cannot maintain Fine Sun Point precision. Therefore, we cannot point the HGA to the required precision. All command and data downlink must be handled via the Omni antennas. After completing the unloading maneuver, STEREO will transition to Coarse Sun Acquisition and proceed to reacquire the science pointing.

Safehold

This mode will be similar the Coarse Sun Acquisition. It will use only CSS and IRU to maintain a power safe mode. This type of attitude determination is too coarse to point the HGA to the required precision, so all commands and data will be routed through the Omni antennas. The software will run from the Attitude Control Electronics (ACE) processor. If possible, the attitude will be maintained with reaction wheels. To maximize the available control authority, the wheels will not be run at a bias speed. If the momentum is too large for the wheels to store, STEREO will transition autonomously to the Delta-H mode for momentum unloading. If STEREO remains in Safehold for more than two weeks, it will be necessary to use the thrusters to unload the momentum. Again, it is recommended to minimize the use of thrusters to minimize risk, fuel usage and contamination concerns on the instrument optics.

6.4.2 Thrusters

The thrusters were sized assuming a SMEX sized bus with a maximum possible torque arm of 0.48 m. The six thrusters are arranged in torque pairs to use for momentum unloading. This arrangement cannot provide any orbit maintenance capability; the science team actually prefers to let the orbits drift slowly apart. However, there will be no way to modify the orbit insertion, a risk the project will have to take. Daily unloading was the driving factor in selecting a thruster size of 0.1 N. Smaller thrust levels will minimize the amount of attitude disturbance produced by the daily Delta-H maneuvers. This will allow the ACS to reestablish science pointing as quickly as possible, minimizing lost science time.



A rough schematic of the thruster system is shown on the left. A thruster force of 0.1 N. The thrusters will interface with the 1553 bus via the attitude control electronics. This system is capable of unloading the daily momentum in approximately 15 seconds. Unloading the maximum tip-off requires a 4 minute burn. There is adequate fuel for three years of momentum unloading, which gives a sufficient margin for a two-year mission life goal.

6.4.3 ACS Issues

Due to external events, certain issues pertaining to the STEREO ACS design could not be resolved in the time frame of this study. These issues and recommendations on potential solutions are presented below.

The IMDC analysis indicates that for the Ithaco Type-A wheels, jitter due to wheel induced flexible mode excitation could easily be of the same order of magnitude as the requirement. The analysis needs to be refined to estimate a jitter profile and the spacecraft's response. This will indicate whether quieter wheels (eg, better balance and isolation of the Ithaco wheels or selection of TRACE-type wheels) will resolve the problem, if an Image Stabilization System (ISS) is necessary, or a combination of both is required.

Because STEREO is a relatively small spacecraft and there are a number of appendages (1 boom, 3 wires, 2 array "wings"), there is concern over the separation between the controller bandwidth and the structural modes. From a controller design standpoint, a separation of a decade between the controller bandwidth and the first free-free structural mode frequency is desired. Again, further refinement of the analysis is required to determine if any trades must be made. Very low control bandwidth is recommended for jitter performance.

6.5 Communications Subsystem

The driver for the design of the STEREO COMM subsystem is the baseline requirement to be able to download 5 Gbits of science data per day (each spacecraft) at the maximum EOL distance of 150 million km (1 AU). The STEREO instrument data rates are shown below in Table 4.

Table 4: STEREO Instrument Data Rates

	Data Rate	Comments
Energetic Particle Detector	200 bps	
Heliospheric Imager	7 kbps	1 image/hr
Magnetometer	200 bps	
Radio Burst Tracker	200 bps	
Solar Coronal Imaging Package	67 kbps	10 images/hr
Solar Wind Plasma Analyzer	200 bps	

Data compression will be done on-board the instrument and is initially baselined as “lossless” or near “lossless” (2:1 or 3:1).

6.5.1 Spacecraft Hardware

The spacecraft bus COMM subsystem is comprised of:

- 1.5 meter, X-band High Gain Antenna (HGA)
- 2 S-band Omni antennas
- 5 watt S-band transponder
- X-band transmitter
- 120 watt, X-band traveling wave tube amplifier (TWTA)

To ensure adequate link margin, a Bit Error Rate (BER) of 10E-5 is specified, with interleaved Reed-Solomon (R/S) and rate ½ convolutional encoding. Ranging is not required due to the ±0.1° attitude control and the 1.7° HGA beamwidth; tracking is available via the S-band transponder. Data is stored in the CDH processor 8 Gbit memory for later playback.

6.5.2 Data Operations Concept

Realtime Science Data is not required and data latency is 36 hours. No redump capability is planned. Insufficient memory is specified to downlink missed passes and previous days data will be overwritten. Generic scheduling will be utilized, as data latency is not an issue.

The DSN 34 meter sub-net will provide primary operations support with the DSN 70 meter sub-net available for emergency and back-up support. Variable data rates will be used to allow for maximizing science data collection, commanding and housekeeping telemetry capabilities.

S-Band telemetry will also be at variable data rates, ranging from 2.2kbps(Omni) on the first month to 2 bps at 1 AU. For the first month (initial check-out), the S-band will be used for real-time housekeeping (HSKP) and playback. When normal operations become routine after the first month, only real-time HSKP will be downlinked on the S-Band.

X-Band telemetry will also be at variable data rates, ranging from 20Mbps at the start of normal operations to 770Kbps at 1 AU using the 34m DSN Antenna. There will be no X-band downlink the first month and any instrument science data available will downlinked using the S-band.

The variable data rates mentioned above are a function of the increasing distance of the STEREO spacecraft from Earth, being closer at the beginning of the mission and moving further away as time progresses (see Figure 2). As the distance increases, link margin decreases; to recover adequate link margin, the data rates might need to be decreased. Earlier in the mission, when closer to earth, excessive link margin allows higher data rates. A higher data rate means that more data could be downloaded if desired; another option would be to download the same amount of data in a shorter time.

From a science standpoint, there may be times when more than the nominal 5 Gbits of data is desirable. From a cost standpoint, less time using the DSN sub-net means lower costs. Figure 9 shows the relationship between:

- ➔ distance and maximum possible data rate (equipment capability)
- ➔ distance and time to dump 5 Gbits at max data rate - holding the data dump constant results in less contact time earlier in the mission, resulting in lower DSN costs
- ➔ distance and maximum data downlinked for a two hour contact - holding the contact time constant results in more data early in the mission, but has higher DSN costs

A combination of the above options can be used to trade between mission costs and maximizing useable science data return. Appendix B compares the telemetry rate versus available communication link margin needed to meet the daily data dump requirement of 5Gbits per day. It should be pointed out that as the distance from earth increases, the round trip time for the signal will be several minutes. This makes it very difficult to do real-time commanding and should be taken into account during the instrument design process.

STEREO to DSN 34m X-band

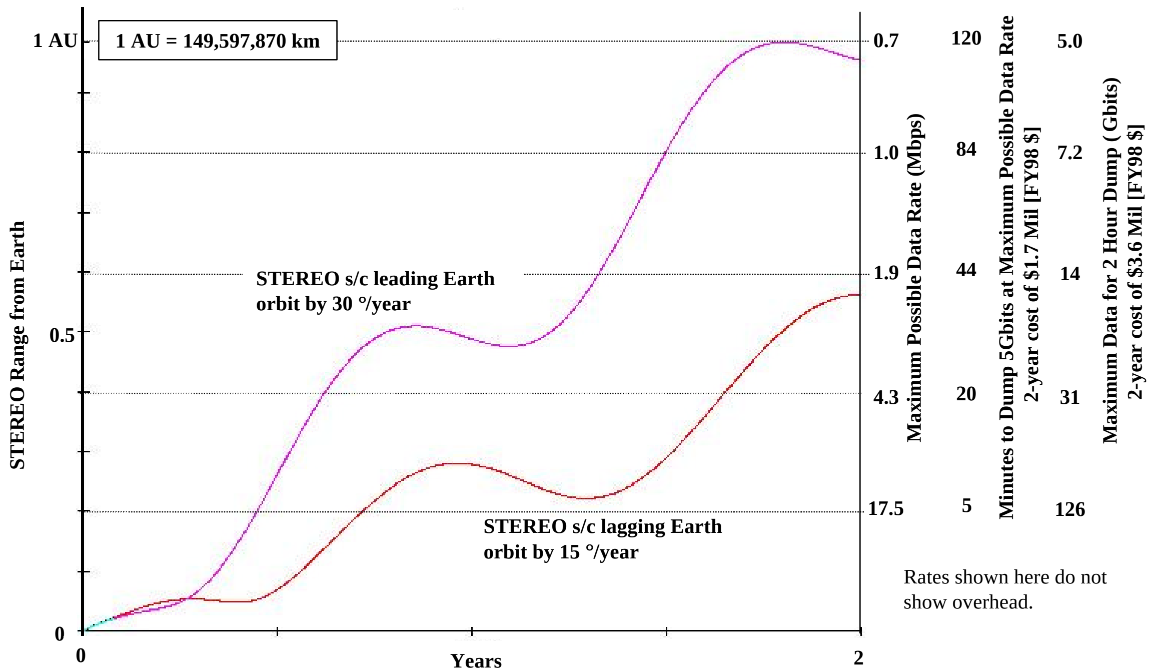


Figure 9: STEREO to DSN 34m X-band

6.6 Software Subsystem

A software development plan will be developed for written for all software defining the development and test methodologies to be used. All software will be written in a commonly used high level language, such as Ada, C or C++.

6.6.1 Flight software will minimally provide the following functions:

- Safehold control (in separate processor from ACS control).
- Real-Time, stored , and event driven commanding capability.
- Telemetry collection and downlink.
- Housekeeping telemetry and science data storage and playback.
- Battery control management
- Spacecraft Fault Detection and correction
- Science Instrument management.

6.6.2 Flight Software Test and Maintenance

A flight software test facility comprised of simulators and engineering models will provide a platform for flight software verification, flight simulations for the operations team, and post-launch flight software maintenance.

Flight software testing will, at a minimum, provide traceability to all flight software requirements in order to ensure that all flight software functionality has been verified.

This facility will also be used for post-launch flight software maintenance activities, which will include the development and verification of all flight software modifications and patches.

6.6.3 Ground Software

Command of the spacecraft and monitoring of spacecraft telemetry during ground testing as well as flight will be provided by the same software. This ensures efficiencies in the development of command software, eliminates potential mismatches of databases between ground and flight, as well as providing many test hours with flight systems. Additional ops control requirements are described in Mission Operations.

7 DESIGN TRADES

As with most designs, there are areas where trades can be made, depending on the needs and goals of the design. In the case of the STEREO design, there are two issues that remain to be addressed:

- The low mass margin (< 10%)
- Final ACS design to meet the jitter requirement

The following paragraphs discuss possible trades.

7.1 Launch Vehicle

It is possible that the mass margin issue could be resolved by selection of an alternative launch vehicle with more payload capability than the ATHENA II selected by this study.

The NASA Shuttle and ARIANNE 5 both possess a payload capability sufficient to provide adequate margin (>30%) for STEREO. Additional study is required to adequately identify interface and other requirements and determine if either vehicle is a viable option. In addition, use of these vehicles would cost considerably less than an ELV.

7.2 Instruments

Since the instruments used in this study are strawman instruments, with the actual instruments selected via the AO process, it is difficult to determine where trades might be made, particularly to help resolve the mass margin issue. It is possible extra consideration could be given to instrument proposals that come in under the strawman mass estimates. Also, the actual instruments proposed could have different requirements and physical configurations that would allow a lighter structure design.

The addition of an Image Stabilization System to the SCIP should be examined as a potential solution to meeting the STEREO jitter requirement.

One area where significant impact could be made with the instruments would be to require higher (lossy) levels of data compression; this could substantially reduce the volume of science data that needs to be downlinked. The impacts of this kind of reduction are discussed in the sections talking about the COMM and CDH areas. This should be looked at carefully to determine if this requirement should be in the instrument AO.

If the instruments that produce large data volumes can buffer their own data, the RS-422 interface could be replaced with the baselined 1553 Bus, or an additional one.

7.3 Spacecraft

If the mass margin cannot be resolved through selection of an alternative launch vehicle, the spacecraft mechanical bus and subsystems can be examined for mass savings: selection of components and subsystems meeting the performance requirements but having less mass; adoption of advanced packaging techniques; reducing performance in return for mass savings. Possible trade areas by subsystem are discussed in the following sections.

7.3.1 Communications

A number of trades can be looked at in the COMM area:

- A higher power amplifier could allow a smaller HGA and gimbal; array size and battery size need to be looked at closely to ensure no offsetting mass growth to handle the increased power requirement; thermal requirement also needs to be considered
- Reduce the science data return at EOL. Possible alternatives are to increase the instrument data compression ratio or directly reduce the daily data dump. Depending on the final compromise, this could lead to reductions in amplifier power, with possible reductions in battery size and array size; HGA size might also be reduced. Significant data rates could still be available early in the mission.
- Increase ground contact time to allow for reduction in the spacecraft communications equipment. This is highly unlikely with the DSN 34 or 70 m antennae, but does appear to be feasible by buying/refurbishing a 26m system. However, this option is 2 to 3 times more costly than using DSN and is therefore not recommended.

7.3.2 CDH

A reduction in the science data return would result in less memory required; this would provide a modest savings in mass, power and cost, depending on the magnitude of the data volume reduction. If the reduction in volume is significant enough, it could reduce the SCIP instrument data rate to the point that the RS422 interface is not required.

7.3.3 ACS

The IMDC has identified trades that can be made against some of the baselined ACS components and modes:

- Lower performance IRU – acquisition and ΔH modes do not require a high performance IRU; a lower power/mass/cost IRU may be appropriate.
- Eliminate CSS control from fine sun acquisition and ΔH modes
- Transition to fine sun acquisition following daily ΔH burn

- PPT vs cold gas – this requires low tip-off rates, which affects kick motor selection
- Hot gas vs. Cold gas

The following trades should be studied to resolve the jitter requirement issue:

- Eliminate wheel speed bias – infrequent jitter violations vs increased daily momentum storage
- Provide accommodation for less wheel excitation through better balance, isolators, for example, or select quieter wheels (eg, TRACE)

8 RECOMMENDATIONS

The following recommendations are offered for the continued formulation of the STEREO Mission:

- ➔ Discussion with DSN scheduling personnel during the IMDC process has given greater confidence that 2 hours of DSN contact time (per spacecraft) could be specified in this design. However, no commitments were given over the phone by JPL and it is **highly recommended that a formal request be submitted to the DSN (through the GSFC SOMO) in the very near future.**
- ➔ Perform additional analysis for the ACS requirements to determine the best solution for meeting the STEREO jitter requirement.
- ➔ Since the end of life data volume requirement specified in this study is such a driver for the COMM subsystem, **careful consideration should be given to specifying a high (lossy) level of data compression for the instruments in the AO.** Since the problem becomes more significant as mission time progresses, a variable, uploadable data compression capability might be suggested (less lossy compression early in the mission, lossier later in the mission).

Appendix A

Ground Data Systems Control Requirements

The Integration and Testing Ground Data System shall be responsible receiving and processing spacecraft data during all phases of spacecraft integration and test. This includes Pre-launch and Launch activities, too. The ;Integration and test GSE shall support the following functions:

Accept S Band CCSDS formatted telemetry downlink at rates up to 440Kbps (encoded).

Accept X Band CCSDS playback telemetry downlink at rates up to 40 Mbps (encoded).

Accept S Band CCSDS transfer frames from S/C and inject into the Launch vehicle data stream (Launch vehicle simulator).

Provide I/O interface to S/C for display of opto messages.

Perform real-time monitoring of housekeeping data at rates up to 64Kbps.

Distribute housekeeping data to individual subsystem GSE"s in real-time.

Distribute Spacecraft , Instrument, and Science Data to Instrument GSE in real time data rates up to 2.25 Mbps.

Archive X-Band playback data and provide distribution capability distribute to Instrument GSE using (ex. TCPIP, FTP).

Archive raw telemetry stream.

Archive formatted spacecraft housekeeping data for playback.

Support CCSDS telecommand format using COP-1 at rates up to 2Kbps.

Accepts command loads from the instruments GSE's for uplink to Spacecraft.

Use Command and Telemetry database compatible with the operational system.

Distribute spacecraft database.

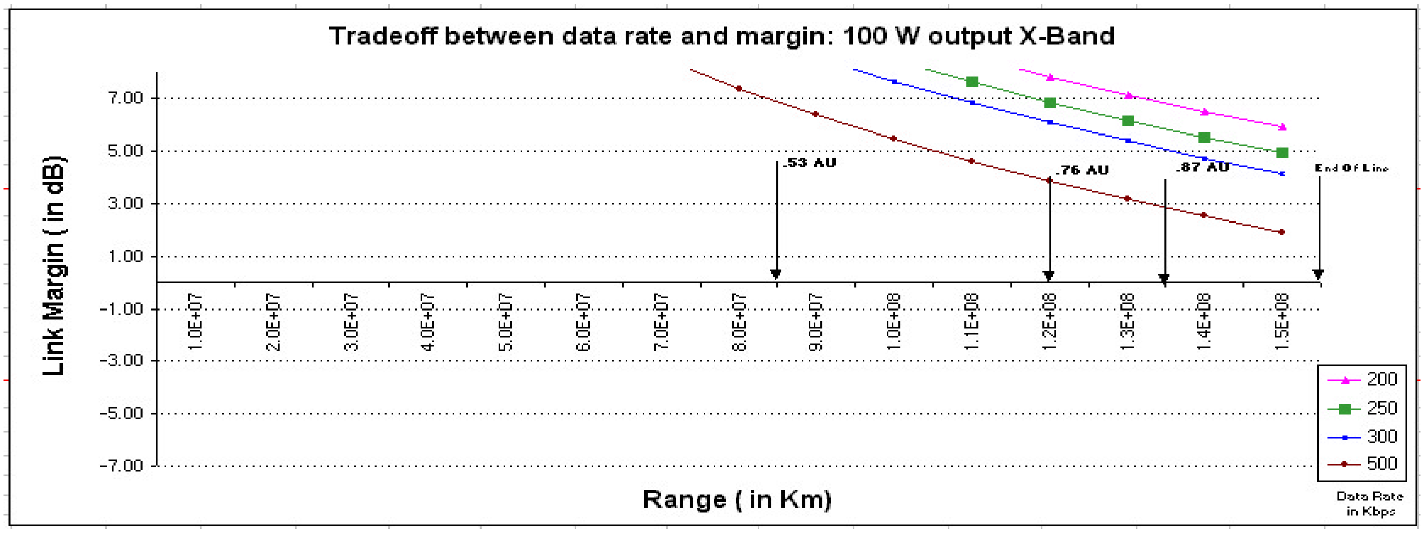
Transmit telemetry transfer frames using the IP protocol

Receive spacecraft commands formatted into NASCOM 4800-bit blocks using the IP protocol.

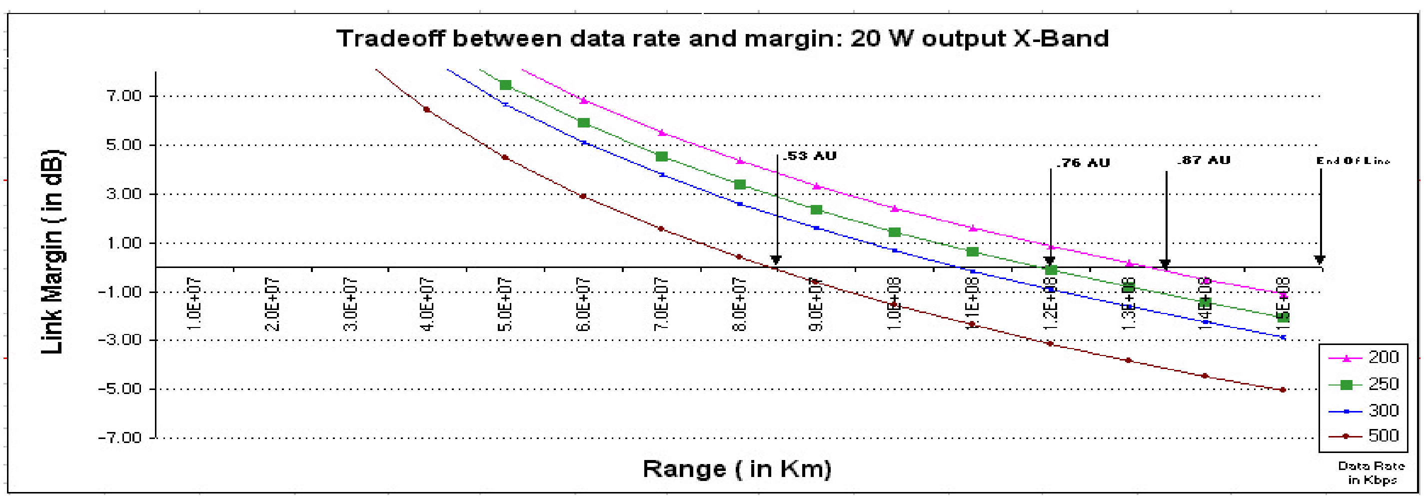
Simulate the DSN interfaces to the operations system for Command, S-Band telemetry and X-Band data streams

Support multi-casting in the IP network

Appendix B



Plot 1- Baseline communication system design with an X-Band transmitter output power of 100Watts. The daily science dump requirement is easily met using a constant telemetry rate of 500 Kbps (takes 2.78 hours to dump the data). Not much improvement is obtained if using the maximum rate dictated by the design approach (it will take 2 hours to complete the dump at 700 Kbps).



Plot 2- A different design approach will be to use variable telemetry rates. Up to the first year (.5 AU) into the mission, there is enough link margin to meet the daily dump requirement using 500 Kbps. The telemetry rate then could be gradually reduced and still meet the daily dump requirements. A significant repercussion is the longer period of time needed to dump all the data. For example, if the telemetry rate is 300 Kbps, it will take almost 5 hours to dump the 5Gbits/day. At 400 Kbps, it will take 4 hours.